CONFIDENTIAL

20. A STUDY OF HYPERSONIC INLET TECHNOLOGY

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SUMMARY

The present state of hypersonic inlet technology is summarized briefly to indicate the need to use a computer program as a design tool. Program usage is described, evaluated, and extended to the design of an inlet for efficient operation between Mach numbers 0 to 5.2. The study shows that the present state of the art does not provide adequate system performance for propulsion systems designed for Mach numbers greater than 3.0 that burn fuel in a subsonic stream. For propulsion systems that burn fuel in a supersonic stream, good progress has been made in attaining adequate inlet performance. It is believed that the performance achieved for most hypersonic inlet systems can be improved by advanced computer programs.

INTRODUCTION

An important objective of hypersonic inlet research has been to understand and predict hypersonic inlet compression flow field phenomena. Available data suggest that accurate analytical means are required to predict inlet performance. The present study uses a recently developed inlet computer program and shows the need for accurately predicting the compression phenomena in developing a feasible inlet design.

The main objectives of this paper are shown in figure 1. The first is to summarize the present state of the art of representative inlet systems up to a Mach number of 8.0 for propulsion systems employing engines that burn fuel in a subsonic stream. A summary is also presented for inlets from Mach numbers of about 4.0 to 15.5 for propulsion systems employing engines that burn fuel in a supersonic stream. The inlets for the former propulsion systems will be termed subsonic burning inlet systems, and the latter, supersonic burning inlet systems. Because the state of the art does not provide adequate system performance, a better approach to the prediction of inlet flow fields is required. The second objective, therefore, is to describe and evaluate a recently developed computer program for viscous inlet flow. Finally, the need for accurately predicting the inlet flow field is demonstrated by using the program to investigate analytically a short axisymmetric inlet system designed for high performance for Mach numbers from 0 to 5.2.

SYMBOLS

$\frac{A_o}{A_c}$	ratio of free-stream tube area to capture area
C_{D_a}	additive drag coefficient
D	capture diameter
$\frac{h}{H}$	ratio of local to total throat height
M	free-stream Mach number
M_e	boundary-layer-edge Mach number
$\frac{p_d}{p_u}$	ratio of downstream to upstream static pressure
P_r	Prandtl number
T_t	total temperature
T_w	wall static temperature
$\frac{u}{u^*}$	ratio of local to throat center-line velocity
δ	boundary-layer thickness
δ^*	boundary-layer displacement thickness
η_{KE}	kinetic energy efficiency

STATE OF THE ART

A number of different types of inlet systems have been proposed for hypersonic vehicles, and wind-tunnel testing of some of these systems has been completed. The three main types for subsonic burning engines along with the advantages and disadvantages of each are shown in figure 2.

The longest inlet, and consequently the heaviest and the one requiring the most cooling, is the internal compression type. It is the most difficult to start. In addition, it requires very high bleed to attain the high potential internal performance theoretically possible at its design Mach number. It does, however, have some advantages as listed in the figure, but overall considerations would not make it as attractive as the two inlets below.

The external compression inlet has several serious faults. Four distinct disadvantages are listed in the figure which may eliminate this type of inlet from consideration for a hypersonic transport. On the positive side, it is the shortest of all inlets and has no starting requirements; that is, no geometry change is required to start the inlet because the terminal shock wave is external to the cowl lip. Inadequate off-design mass flow, however, creates the need for a contracting centerbody as indicated by the dashed lines.

The mixed compression inlet has high potential internal performance but the full potential of this type of inlet apparently has not yet been achieved. For a wraparound turboramjet engine the off-design mass flow required can be achieved with this type of inlet through a contracting centerbody system, but translation of the centerbody provides only a marginal or unacceptable mass flow. As can be seen by comparison with the external compression inlet above, the mixed compression inlet is somewhat longer, therefore heavier, and requires more cooling at hypersonic speeds. Accurate analytical tools should allow design of a minimum length mixed compression system which will have adequate performance.

Experimental performance of the three systems is indicated in figure 3. Engine-face pressure recovery is plotted as a function of inlet Mach number. The dashed line shows for comparison a reasonable goal for the pressure recovery and was derived from recent hypersonic transport studies (ref. 1). Up to $M = 4.0$ adequate performance appears to be possible. Beyond $M = 4.0$ (ref. 2) the performance is marginal when compared to the transport goal. Recent work with axisymmetric inlet systems is shown by the light line up to $M = 3.0$ (ref. 3). Extrapolation of this work to higher Mach numbers (ref. 4) indicates what might be attained with more research. The external compression system (ref. 5), indicated by the heavy line, appears competitive in the higher Mach number range. The single point shown for the all internal contraction inlet system (ref. 6) indicates high recovery, but at the expense of about 19-percent boundary-layer bleed. It should be emphasized that the performance data shown are composites of only the best performance attained for each of the inlet types, and no single inlet has achieved the hypersonic transport goal over the complete Mach number range. From the foregoing discussion it appears that for subsonic burning most of the advantages lie with the mixed compression inlet if the potential performance can be achieved experimentally over the complete Mach number range.

For supersonic burning inlet systems there appears to be more freedom in design. Figure 4 shows several inlets that have been or are being tested. The first inlet is one currently being readied for the NASA Hypersonic Research Engine for Mach numbers up to 8.0. It is an axisymmetric inlet designed so that forward translation of the centerbody will close off the inlet flow to reduce the cooling load during nonoperating conditions on the X-15. The second inlet is an axisymmetric design currently being tested up to $M = 10.0$ and is being considered for use in a self-accelerating vehicle. The three remaining designs appear less conventional having fixed geometry with self-starting capabilities down to lower Mach numbers. The difficulty of analyzing the flow fields is considerably greater than for the axisymmetric inlets. The third inlet is derived from an axisymmetric nozzle design.

Conventional means can be used to analyze this inlet although they do not predict the exact flow field. The inlet is currently considered for missile applications. The fourth and fifth inlets have more complicated three-dimensional flow fields requiring analytical means which are not fully developed as yet. The fifth inlet differs from the others in that combustion is initiated in the throat causing thermal compression of the flow. This allows self-starting to lower Mach numbers than the third or fourth inlets and it may be shorter entailing savings in weight and cooling requirements.

Inlets for supersonic burning must be compared on a different basis from that for subsonic burning. For supersonic burning, measurements of performance are usually made in the throat region of the inlet, and the kinetic energy efficiency is generally used to describe the performance. Figure 5 shows a band of experimental performance (refs. 7-16) for inlets such as just described. Pressure recovery at the throat is plotted as a function of Mach number. The dashed lines are for constant kinetic energy efficiencies of 98 and 96 percent. Performance falling above or between these lines is considered adequate. It is evident that good progress has been made in attaining adequate performance. The broader problem lies in combining the inlets with combustors and exit nozzles so as to achieve adequate system performance over the flight profile.

COMPUTER PROGRAMS

Turning now to the second objective, sophisticated tools in the form of computer programs will be described and evaluated. Two computer programs have been used in the present study which employ the method of characteristics. One program computes only the inviscid portion of the flow and will be termed the inviscid program (ref. 17). This program requires only about 2 minutes of computer time on an IBM 7094. The other program couples simultaneously the inviscid real gas solution with a boundary-layer solution and will be termed the viscous program. The program requires one-half to one hour per solution; hence, much computer time is saved when the inviscid program is used for preliminary designs. The viscous program was developed under NASA contract with Lockheed-California Company and is described and evaluated in reference 18. The capability of the viscous computer program is as follows (see fig. 6): From a known blunt-body solution at the nose, the program will compute the boundary layer under the blunt nose shock layer followed by computation of the real gas inviscid flow field simultaneously with the boundary layer. It will calculate a blunt cowl lip solution using the local upstream conditions. The program then calculates the internal flow field including shock-wave—boundary-layer interactions. If the bow shock wave falls inside the cowl lip as shown in figure 6, the program calculates the crossed shock waves and the associated vortex sheets. The theoretical approaches used in the program are believed to be valid, but further work, which will be described in the succeeding papers, may yield improved theories for flow details such as boundary-layer—shock-wave interactions. Such improvements are expected to be incorporated in the viscous program eventually.

The usefulness of the viscous program has been established by comparison with wind-tunnel results. Figure 7 is a comparison of computed and experimental velocity profiles across the throats of two different inlets. The Mach number 4.0 and 3.0 mixed compression designs shown have sharp noses and lips. Since the experimental profiles for both of these inlets were measured while some of the boundary layer was removed, one might expect an experimental profile with a thinner boundary layer than that predicted. However, a thicker boundary layer was observed on the centerbody for both inlets. Even though the comparisons may not agree well, the theories used are believed to be as good as is currently available. Better agreement should be achieved in the future with the development of more accurate theories.

PROGRAM USAGE

Proceeding to the final objective, the program usage will be extended to the design of an inlet. In the hypersonic Mach number range from 5.0 to 8.0, wraparound turbofan or turboramjet engines appear attractive and require inlet systems with subsonic as well as supersonic diffusion. Figure 8 shows such a system designed for Mach number 5.2 mated to a wraparound turbofan-ramjet. The axisymmetric inlet system has a mixed compression supersonic diffuser and a rather short subsonic diffuser. The propulsion system is about 3.0 capture diameters long from the cowl lip to the nozzle exit. The inlet system is 1.25 capture diameters from the cowl lip to the engine face. The dashed lines represent sectional views of two off-design modes of varying the geometry. The upper sectional view shows the centerbody translated to the Mach number 1.0 position while the lower view shows the centerbody in a contracted position for Mach number 1.0. The mass flow is only 20 percent of the capture area mass flow for the translating centerbody version, but is 50 percent for the contracting centerbody. Since for best performance candidate wraparound turbofan or turboramjet engines require on the order of 50 percent of more capture mass flow at $M = 1.0$, the contracting centerbody is an attractive mode of off-design operation for a mixed compression axisymmetric inlet system.

At $M = 5.2$ the subsonic diffuser can be rather short, mainly because the supersonic diffuser contraction ratio is high. That is, because the Mach number in the throat region after the terminal shock-wave system is about 0.5 to 0.7 and the throat height is small, the area ratio to diffuse the flow to $M = 0.2$ or less can be achieved in a relatively short axial distance with low diffusion efficiency losses. Beyond this point the sudden expansion losses are quite small since the Mach number is low. Turning the flow into the wraparound ramjet should also cause only small flow diffusion losses since the Mach number is only about 0.15 into the ramjet. The shapes of the subsonic diffusers for both the translating and contracting centerbody versions at Mach number 1.0 do not appear to offer a diffusion loss problem since tests of similar shapes have indicated that the losses are small.

Figure 9 shows the details of the internal supersonic flow field of one proposed design as predicted by the computer programs. Only the contours in the region of the cowl lip and throat are shown for clarity. The internal shock-wave system predicted by the viscous program is shown by the wavy lines.

For comparison, the system predicted by the inviscid program is shown by the solid lines. It is evident that the inviscid program predicted a shock-wave system which extended over a greater length than that predicted with the viscous program. This is mainly because boundary-layer displacement thickness shown by the dashed lines was not taken into account. The pressure ratios across each impingement predicted with the inviscid and viscous programs are indicated in the table at the bottom of the figure. It is evident that pressure ratios predicted with the inviscid program are considerably lower than those predicted with the viscous program again mainly because boundary-layer displacement thickness was not taken into account. The accuracy of the values shown for the viscous program depends to a considerable extent on the accuracy of the boundary-layer and boundary-layer—shock-wave interaction theories used in the program. These problem areas are discussed in this conference by Mr. Gnos and by Mr. Watson.

It is believed that separation of the boundary layer must be avoided both for on and off design if high performance is to be achieved. The hypersonic program has predicted the approaching boundary-layer thickness and Reynolds number for each shock-wave impingement shown in figure 9. Knowing these quantities the incipient pressure rise for separation can be estimated. Figure 10 is a plot of pressure ratio as a function of local boundary-layer edge Mach number ahead of the impingements. The dark band is an envelope of experimental data for incipient separation of a turbulent boundary layer caused by a shock wave impinging on a flat plate (ref. 19) corresponding to the range of Reynolds numbers based on the boundary-layer heights ahead of each impingement. Computed pressure ratios that fall above the band should separate; those that fall below should not separate. The inlet, however, has curved surfaces with high local pressure gradients in the region of the impingements. What part this may play in the accuracy of predicting separation is not known. The pressure ratios listed in the table of figure 9 are plotted in this figure. Those ratios predicted by the inviscid program shown by the filled symbols do not indicate separation, but these values, as previously indicated, are overly optimistic. It appears that more refinement in the design of the inlet contours is required since the pressure rise predicted with the viscous program for the second impingement falls in the region of separation.

The effect of cooling the boundary layer can also be predicted by the viscous program. Figure 11 shows the effect of varying the wall temperature ratio. The ratio of boundary-layer displacement thickness to the thickness at the near average adiabatic temperature of 904° R is plotted as a function of the ratio of wall temperature to 904° R for each impingement at $M = 5.2$. As expected, the boundary layer becomes thinner with increased cooling. In addition, cooling has more effect on thinning the boundary layer in the throat than forward on the centerbody as evidenced by comparison of the curves for the first and third impingements. The calculations shown here are for a suitable wind-tunnel total temperature of 1200° R. Actual flight conditions will demand that the walls be cooled to about 0.4 of the near adiabatic wall temperature ratio. Reducing the wall temperature ratio to 0.4 for the 1200° R case reduces the boundary-layer displacement thickness ahead of the first impingement about 20 percent while the thickness is reduced about 30 percent on the succeeding impingements. Cooling is expected to be favorable to the

performance by increasing the incipient pressure rise for separation and increasing the relative area of inviscid core flow in the throat. This assumes, however, that the heat removed by cooling is accomplished with the fuel and can be recovered in the propulsion system when the fuel is injected in the combustor.

CONCLUDING REMARKS

The foregoing study of hypersonic inlet technology has shown that the present state of the art does not provide what is considered adequate system performance over the complete Mach number range for subsonic burning inlet systems. Mixed compression inlet systems appear more promising than either the external or internal compression types for self-accelerating vehicles such as the hypersonic transport. For supersonic burning systems good progress has been made in attaining adequate inlet performance. The broader problem lies with attaining adequate performance over the flight profile when an inlet, a combustor, and exit nozzle are combined. It is believed that improved performance can be achieved for most hypersonic inlet systems through use of advanced computer programs which can accurately predict hypersonic inlet flow fields.

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STUDY OF HYPERSONIC INLET TECHNOLOGY
OBJECTIVES

- SUMMARIZE PRESENT STATE OF THE ART
- DESCRIBE AND EVALUATE COMPUTER PROGRAM
- EXTEND PROGRAM USAGE TO INLET DESIGN

Figure 1

INLET TYPES
SUBSONIC BURNING

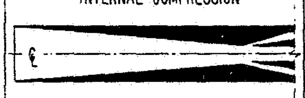
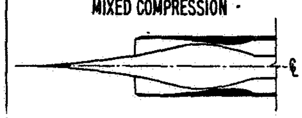
ADVANTAGES	TYPES	DISADVANTAGES
1. HIGH P.R. POTENTIAL 2. LOW COWL DRAG 3. LOW C_{D_a} 4. HIGH MASS FLOW	INTERNAL COMPRESSION 	1. LONGEST INLET 2. DIFFICULT TO START 3. HIGH B.L. BLEED
1. SHORT INLET 2. NO STARTING REQUIREMENT	EXTERNAL COMPRESSION 	1. HIGH COWL DRAG 2. HIGH C_{D_a} 3. MARGINAL P.R. 4. INADEQUATE TRANSONIC MASS FLOW
1. HIGH P.R. POTENTIAL 2. LOW COWL DRAG 3. LOW C_{D_a}	MIXED COMPRESSION 	1. POTENTIAL UNREALIZED 2. MARGINAL TRANSONIC MASS FLOW 3. LONGER THAN EXTERNAL COMPRESSION INLET

Figure 2

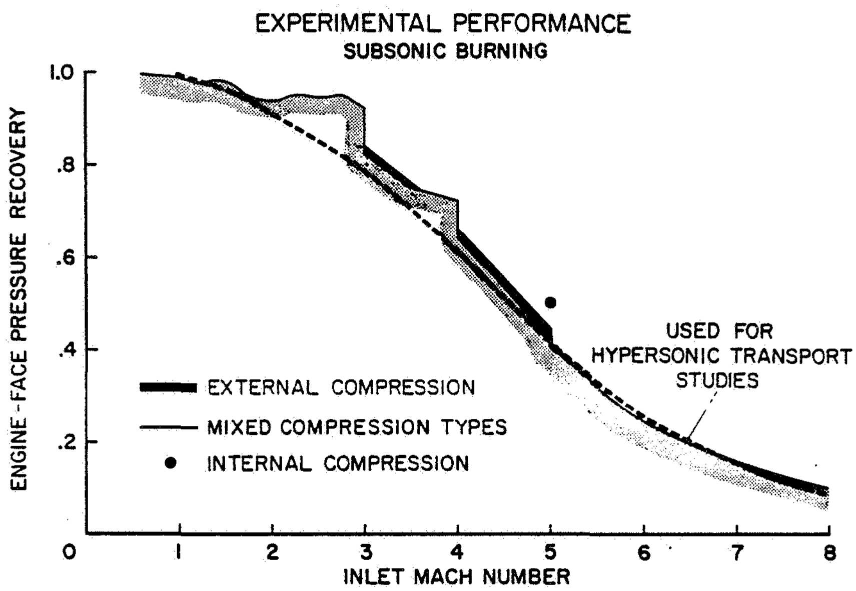


Figure 3

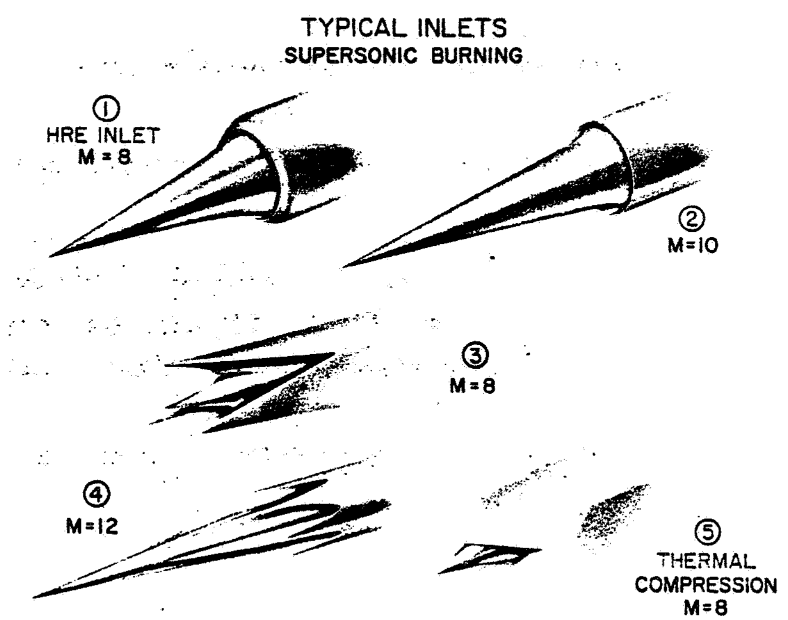


Figure 4 AAA375-4

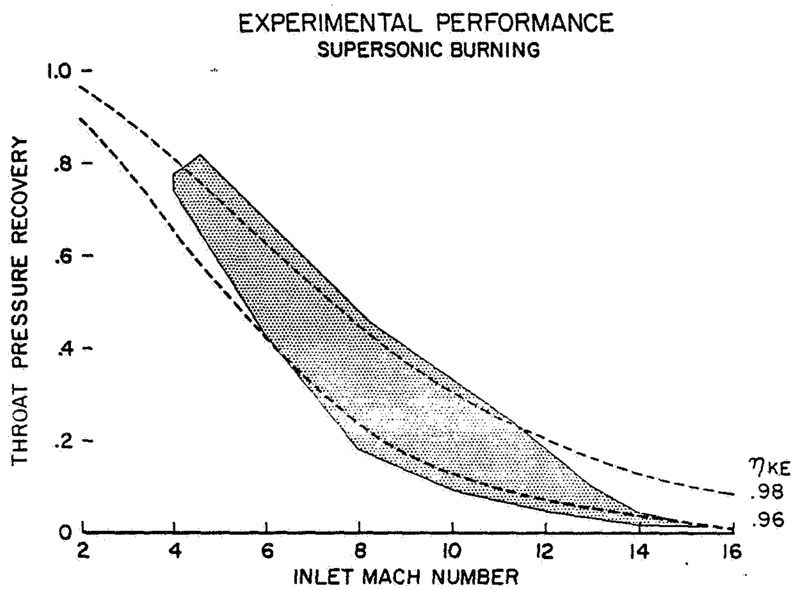


Figure 5

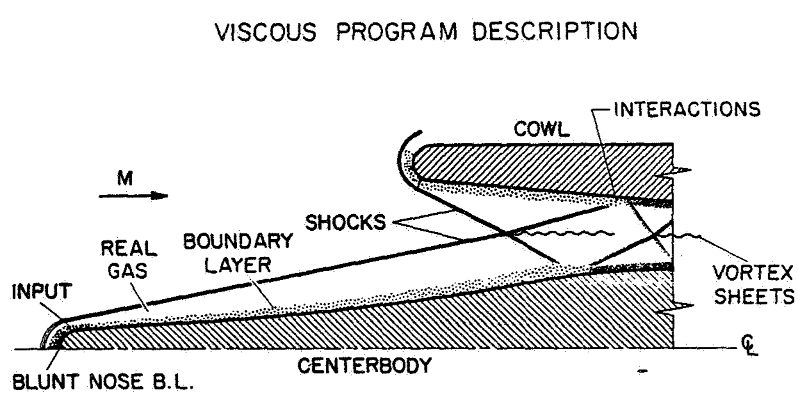


Figure 6

THROAT FLOW PROFILES

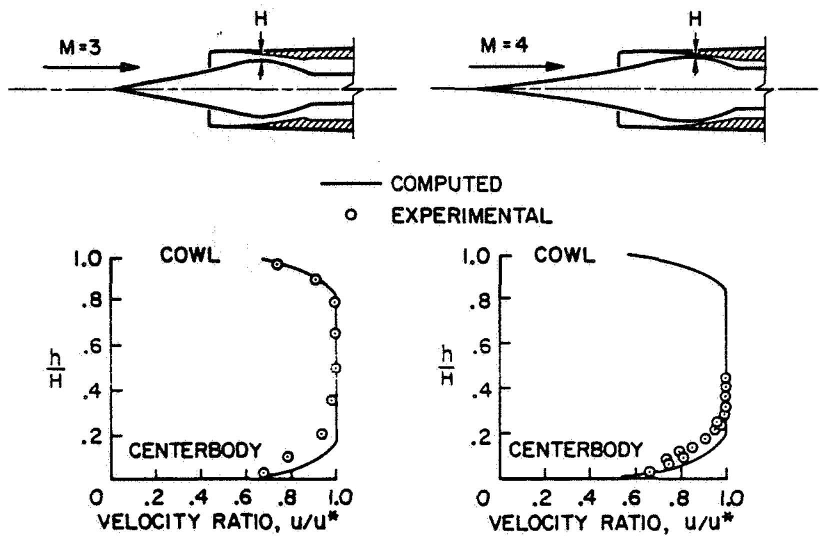


Figure 7

MACH 5.2 AXISYMMETRIC INLET SYSTEM

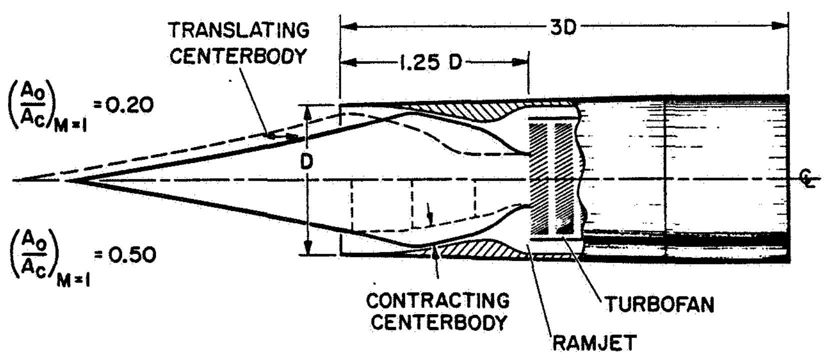
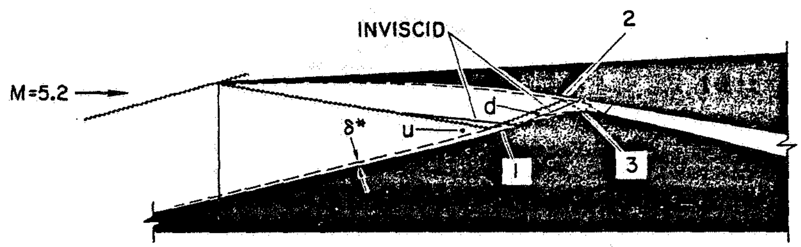


Figure 8

FLOW FIELD MODEL
COMPUTER PROGRAMS



$\frac{p_d}{p_u}$	1	2	3
INVISCID	2.37	1.59	1.62
VISCOUS	4.80	3.74	1.88

Figure 9

BOUNDARY-LAYER SEPARATION CRITERIA
M=5.2

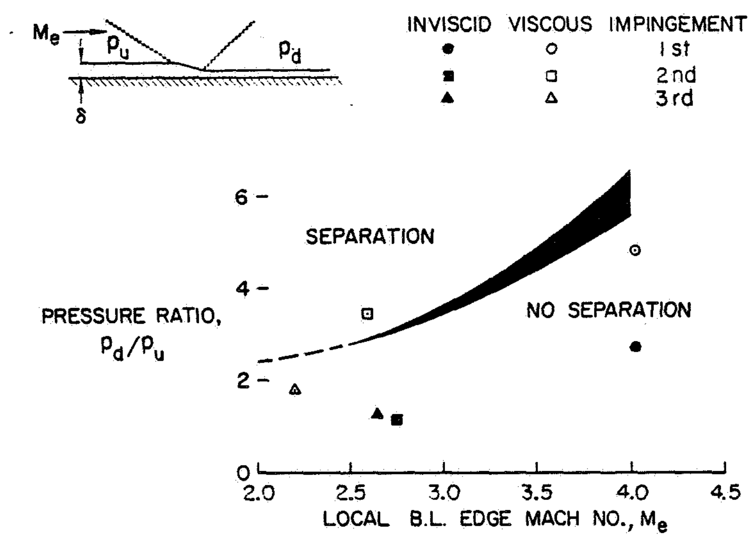


Figure 10

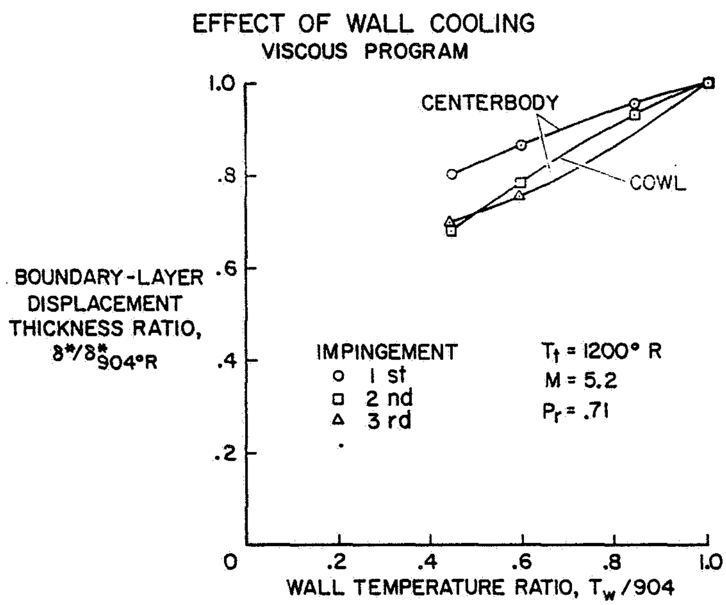

EFFECT OF WALL COOLING
VISCOUS PROGRAM
CENTERBODY
COWL
BOUNDARY-LAYER DISPLACEMENT THICKNESS RATIO, δ*/δ*904°R
IMPINGEMENT
○ 1 st
□ 2 nd
△ 3 rd
Tt = 1200° R
M = 5.2
Pr = .71
1.0
.8
.6
.4
.2
0
.2
.4
.6
.8
1.0
WALL TEMPERATURE RATIO, Tw/904

Figure 11